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# TECHNICAL MEMORANDUM

## X-149

# PERFORMANCE SUMMARY AND ANALYSIS OF A MACH 3.0 DESIGN AXISYMMETRIC ALL-EXTERNAL-COMPRESSION DOUBLE-CONE INLET FROM MACH NUMBER 3.0 TO 0.8


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# PERFORMANCE SUMMARY AND ANALYSIS OF A MACH 3.0 DESIGN AXISYMMETRIC ALL-EXTERNAL-COMPRESSION DOUBLE-CONE INLET FROM MACH NUMBER 3.0 TO 0.8*

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## SUMMARY

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Pressure recovery, mass-flow ratio, and coefficients of cowl pressure and additive drag are summarized for an all-external-compression double-cone inlet through the Mach number range of 3.0 (design point) to 0.8.

By translating the spike so that the second oblique shock remained on the cowl lip, the airflow at critical conditions of the inlet closely matched that required by a turbojet between Mach numbers 3.0 and 2.4. At lower Mach numbers the throat was choked, which reduced the captured mass flow and increased normal-shock additive drag. The total drag reached a peak value at about Mach number 1.3 of 3.4 to 3.7 times the Mach number 3.0 value. Because of the reduced mass flow due to throat choking, oversizing of the inlet was required in order to avoid supercritical matching, and at higher Mach numbers a bypass drag for the excess captured flow was accepted. This procedure at best produced a relatively flat effective-thrust curve in the choked-throat region, with a small bypass-drag penalty at the design point.

By using a fixed ram-scoop boundary-layer bleed, the effective thrust was increased 20 percent at Mach number 3.0 and was better than the no-bleed inlet down to Mach number 2.0. At Mach numbers between 2.0 and 1.4 the bleed-system mass flow increased without any appreciable pressure-recovery benefit and hence required a relatively larger inlet with the attendant oversizing penalties.

## INTRODUCTION

Efficient, compatible operation of the air inlet and turbojet engine throughout the Mach number range has become both more necessary and difficult as the design Mach number has been raised to 3.0. Many different

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inlet designs have been suggested. One of these is the conventional external-compression double-cone inlet. This type of inlet has been extensively investigated in references 1 to 4 over a range of Mach numbers from the design point of 3.0 to 0.79.

References 1 and 2 present the inlet performance from Mach numbers of 3.0 to 1.98 as determined in the NASA Lewis 10- by 10-foot supersonic tunnel. The best configuration (20% projected cowl area having a diam. of 17 in.) of references 1 and 2, with and without a centerbody ram-scoop boundary-layer bleed, was tested in the NASA Lewis 8- by 6-foot supersonic tunnel over the Mach number range of 2.07 to 1.48 as reported in reference 3. Results obtained on a geometrically similar model, 13 inches in diameter (with a partially modified subsonic diffuser) investigated in the transonic section of the NASA Lewis 8- by 6-foot supersonic tunnel are reported in reference 4 for Mach numbers 1.48, 1.28, 1.00, and 0.79. This report summarizes the pertinent data of references 1 to 4, and the significance of the basic inlet performance is illustrated by means of an inlet-engine matching analysis utilizing a current Mach number 3.0 turbojet engine and a general flight plan.

## SYMBOLS

| | |
|---|---|
| $A$ | area, sq ft |
| $A_{cf}$ | compressor-face or tip area |
| $A_{in}$ | inlet capture area |
| $A_{max}$ | maximum projected area of model |
| $A_t$ | throat area |
| $A_3$ | diffuser-exit flow area |
| $C_D$ | drag coefficient, $D/q_0 A_{max} = C_{D,e} - C_{D,f}$ |
| $C_{D,a}$ | additive-drag coefficient, $C_D - C_{D,c}$ |
| $C_{D,c}$ | cowl pressure-drag coefficient, $\frac{1}{y_{max}^2} \int_{y_l}^{y_{max}} C_p (dy)^2$ |
| $C_{D,f}$ | friction-drag coefficient |
| $C_p$ | static-pressure coefficient, $(p - p_0)/q_0$ |

| | |
|---|---|
| $D$ | drag |
| $F$ | net thrust at operating pressure recovery |
| $\frac{F - D}{F_i}$ | effective-thrust ratio |
| $F_i$ | ideal net thrust at 100-percent pressure recovery |
| $M$ | Mach number |
| $m_b/m_0$ | bleed mass-flow ratio |
| $m_0$ | mass flow in a capture-area stream tube, $\rho_0 V_0 A_{in}$ |
| $m_3/m_0$ | mass-flow ratio, $\rho_3 V_3 A_3/\rho_0 V_0 A_{in}$ |
| $P$ | total pressure |
| $\overline{P}_3/P_0$ | total-pressure recovery |
| $p$ | static pressure |
| $q$ | dynamic pressure |
| $V$ | velocity |
| $w$ | weight flow, lb/sec |
| $\frac{w\sqrt{\theta}}{\delta A}$ | corrected weight flow per unit area |
| $y$ | distance normal to axis of symmetry |
| $\delta$ | ratio of total pressure to NASA standard sea-level pressure of 2116 lb/sq ft |
| $\theta$ | ratio of total temperature to NASA standard sea-level temperature of 518.7° R |
| $\theta_l$ | spike-position parameter, angle between axis of symmetry and line from spike tip to cowl leading edge, deg |
| $\rho$ | density of air |

Subscripts:

| | |
|---|---|
| e | external |

$l$ lip

max maximum

0 free-stream conditions

3 diffuser-exit conditions

Superscript:

$\overline{\phantom{x}}$ area-weighted value

## METHOD OF ANALYSIS

The basic double-cone model (fig. 1), with compression surfaces of $20^\circ$ and $35^\circ$, respectively, and a cowl-lip projected area of 20 percent of maximum area, was designed for both shocks on lip at Mach 3.0. The configuration shown employed a centerbody ram-scoop boundary-layer bleed. The performance of this bleed configuration was found (ref. 2) to be very sensitive to spike position; therefore, the spike-translation schedule, which normally kept the second oblique shock on lip until shock detachment or throat choking occurred, was modified in order to obtain the best performance. The spike schedule modification along with the identification of the selected data from references 1 to 4 is itemized in the following table:

| $M_0$ | Spike-position parameter, $\theta_l$, deg | | Reference ( ) and model identity therein | |
|---|---|---|---|---|
| | No bleed | Bleed | No Bleed | With boundary-layer bleed |
| 3.01 | 29.52 | 29.30 | (1) 20 Percent cowl | (2) Inlet I with ram scoop 2 |
| 2.73 | 29.96 | 29.92 | ↓ | ↓ |
| 2.44 | 30.79 | 30.5 | ↓ | ↓ |
| 1.98 | 31.55 | 31.55 | (3) 20-35 | (3) 20-35B |
| 1.78 | ↓ | ↓ | ↓ | ↓ |
| 1.48 | ↓ | ↓ | ↓ | ↓ |
| 1.28 | 32.1 | 32.1 | (4) 20 Percent cowl | None |
| 1.00 | ↓ | ↓ | ↓ | ↓ |
| .79 | ↓ | ↓ | ↓ | ↓ |

Corresponding models of references 1, 2, and 3 were identical. The smaller transonic model tested in reference 4 was very nearly geometrically similar to the larger model with the major exception of having a much shorter subsonic diffuser aft of the centerbody telescoping joint. A comparison of capture, maximum, and throat areas is given in the following table:

| Model | $A_{in}/A_{max}$ | $A_t/A_{in}(\theta_l = 31.55^\circ)$ | $A_t/A_{in}(\theta_l = 29.5^\circ)$ |
|---|---|---|---|
| Refs. 1, 2, and 3 | 0.797 | 0.478 | 0.361 |
| Ref. 4 | .804 | .478 | .385 |

In the data comparisons, the friction drag has been removed from the total drag because of the difference in external skin area between the two models. The transonic model (ref. 4) was not tested with the ram-scoop boundary-layer bleed.

For a study of the inlet performance over the Mach number range, a matching analysis was made with a current turbojet engine for which an airflow schedule and flight plan are shown in figure 2.

The effective-thrust ratio was chosen as the figure of merit for comparison. It is defined as the ratio of operating net thrust minus the sum of cowl pressure and spillage drags to the ideal net thrust at 100-percent total-pressure recovery with no drag.

As part of the matching analysis, the inlet was assumed to be sized at Mach numbers of 3.0, 1.48, and 1.00. Below-design Mach number sizings result in an excess of inlet capacity over part of the flight path. Bypassing of the excess flow assures inlet operation at or near the critical flow point. For the inlet having bleed, the operating point was chosen, in general, to be slightly subcritical at some Mach numbers in order to utilize the higher pressure recovery available for small bow-shock spillages.

The bypass-drag penalty was computed on the optimistic basis of an axial discharge of the excess flow fully expanded at engine-face total-pressure recovery. The bleed-flow drag was evaluated in this same manner, also optimistically, since the bleed-flow recovery is lower. Bleed mass-flow ratio was estimated from the difference between supercritical mass-flow ratios for configurations with and without the ram-scoop bleed and assumed to have the same value at the match point.

The difference in friction drag between the largest and smallest of the inlet sizings used for the matching analysis was calculated to be less than 0.5 percent of the ideal net thrust and hence was neglected.

Theoretical mass-flow ratios were computed from two Mach number regions for both the fixed and translating spikes. For the Mach numbers up to 2.0, a choked-throat calculation was made using an assumed throat total-pressure recovery.

For Mach numbers greater than 2.1, two conditions are possible. First, when the schedule of spike translation maintains the second shock on the cowl lip, the theoretical mass-flow ratio can be found by treating the problem as an off-design single cone, since there is no spillage behind the second shock. For the fixed spike position, the second shock does not intersect the lip. The procedure in this case was to follow a capture streamline from the cowl lip to the second shock using a constant-area annulus normal to the second cone. Then the intersection point of the streamline and second-cone shock was treated as the cowl-lip location for an off-design single cone.

## RESULTS AND DISCUSSION

### Inlet Performance

The performance of the fixed- and translating-spike inlets without bleed is summarized in figure 3. This summary includes component drags, total-pressure recoveries, and a comparison of measured mass-flow ratios with theoretical values based on the assumed throat recoveries shown in the figure.

The theoretical mass-flow ratios predict the trend and are in good agreement with the experimental values, considering that a full stream tube was not captured at Mach number 3.0. For the fixed spike the discontinuity in the theory at $M_0 = 1.5$ occurs because of the difference in throat areas (at inlet entrance) between the two models.

As the spike is retracted, throat area increases; and for Mach numbers below 2.1 excessive internal contraction occurs for spike positions greater than 31.55° because of the geometrical relations of the cowl and centerbody. However, the throat area can never exceed that of the telescoping joint station, and this value was used in the theoretical choked-throat calculation for the retracted spike. The actual throat area is probably somewhat smaller. The only method of providing for increased throat area and decreased contraction is to increase the flow area at the slip-joint station by allowing the percent of lip area to increase. As shown in reference 1, this procedure markedly increases cowl pressure drag.

The total drag (actually $C_{D,c} + C_{D,a}$) increases with decreasing Mach number until it reaches a peak value in the region of Mach number 1.3 of about 3.4 to 3.7 times the Mach number 3.0 value. The total drag

is decreased at a subsonic Mach number of 0.8 to about 0.6 of the design Mach 3.0 value. The primary reason for the high peak value of the drag curves is the throat restriction that requires the spillage of air by means of a bow shock. For the fixed spike position the drag coefficient is slightly higher than that of the translating-spike configuration throughout the Mach number range, because more mass flow must be spilled when the second oblique shock does not intersect the cowl lip. Cowl pressure drags are nearly identical for both the fixed- and translating-spike conditions.

Performance (only available between Mach numbers 3.0 and 1.48) of the inlet with the ram-scoop bleed is also shown in figure 3. Total-pressure recovery was appreciably increased at Mach number 3.0 by the ram-scoop bleed but only when the spike was extended and conical shock spillage increased (ref. 2). The effectiveness of the bleed diminished with decreasing Mach number. The bleed mass-flow ratio $m_b/m_0$ varied from approximately 0.030 to 0.08 for Mach numbers 3.0 to 1.48.

The critical total-pressure distortions measured at the compressor-face station (4.5 inlet diam. aft of cowl lip) varied from 2.5 to 4.0 percent between Mach numbers of 3.0 and 0.8 for the fixed-spike, no-bleed configuration. For the variable-geometry inlet, the distortion varied from 2.5 percent at Mach number 3.0 to a peak of 14 percent at Mach 1.98 and decreased to 10 percent at Mach 0.8.

## Turbojet Matching Analysis

Engine and inlet airflows. - Figure 4 shows a comparison of the engine corrected airflow schedule with that provided by three inlet sizings with and without throat bleed. The corrected weight flow per unit area demanded by the engine decreases from about 26 to 15 pounds per second per square foot between Mach numbers 1.0 and 3.0. If the inlet is sized to match the weight flow demanded by the engine at Mach number 3.0 and the spike position is fixed at the design value, the inlet cannot provide the required weight flow at lower speeds at critical flow. Thus, supercritical inlet operation would be necessary, and the attendant lower pressure recovery would greatly reduce the net thrust. In the Mach number region of 3.0 to 2.44, where the second oblique shock can be kept on the cowl lip by retracting the spike, the critical flow capacity of the inlet very nearly equals that of the engine. At lower Mach numbers, however, the capacity of the inlet is grossly restricted because of the choked-throat mass-flow limitations.

If the inlet is made much larger so that critical-flow matching of the inlet and engine weight flows occurs at a lower Mach number such as 1.48 or 1.0, then the inlet capacity is much larger than required by the engine in the higher Mach number range. The weight flow in excess of

engine requirements must be disposed of with the least possible thrust penalty. The least desirable method for disposing of excess flow is that of subcritical operation because of the attendant normal-shock spillage drag and possible invasion of the inlet buzz region. A more efficient method is to operate the inlet at or near critical flow and dispose of the excess flow by means of a bypass discharge or bypass to secondary-air systems such as an exhaust-nozzle ejector (refs. 5 and 6). Excessive inlet flow may be spilled by spike translation in certain parts of the Mach number range without a loss in total-pressure recovery.

At critical flow the lower mass-flow ratio and higher recovery of the bleed inlet provided a weight flow 20 percent smaller than the no-bleed inlet at Mach number 3.0. Thus, when sized for the same engine at Mach 3.0, the bleed inlet, being 20 percent larger, provides a slight excess of inlet capacity down to about Mach number 2.0. However, at lower Mach numbers the capacity of the Mach 3.0 matched inlet decreases markedly when slightly subcritical operation is chosen. The magnitude of this effect is clearly seen by comparing the airflow schedules at Mach 1.48 for critical and subcritical inlet sizings.

Effective-thrust comparisons. - The importance of the preceding general considerations is shown in figure 5 in terms of effective-thrust ratio, which is the ratio of net thrust minus cowl, additive, and bypass drags (neglecting friction) to ideal net thrust. The losses in effective thrust due to drag and pressure recovery are shown individually. Results of sizing the inlet to match the engine are shown for Mach numbers 3.0, 1.48, and 1.0; a bypass system was used to discharge excess and bleed air.

When the inlet is sized at Mach 3.0, the effective-thrust ratio is seriously reduced in the Mach number region below 2.4 for the no-bleed case (fig. 5(a)) because of supercritical operation and drag buildup due to increasing critical additive drag. Increasing the size of the inlet, and thereby avoiding supercritical operation, materially improves the effective-thrust ratio in the Mach number range below 2.4 for the no-bleed case. This improvement is due primarily to the better pressure recovery, since the increase in drag associated with the larger inlet (including bypass drag) was no more than one-third the thrust gain due to pressure recovery.

The improvement obtained by using the bleed inlet is between 9 and 11 percent of ideal thrust at the design Mach number (depending on the size). Furthermore, the bleed inlet when sized on design (fig. 5(b)) shows an advantage over all of the no-bleed inlets down to Mach number 2.0. At lower Mach numbers the Mach 1.00 and 1.48 no-bleed sizes show the higher effective-thrust ratios. In order to improve even marginally the effective thrust below Mach 1.8, an increase in the inlet size was necessary. In fact, the required size increase was large enough to reduce

the effective thrust materially above Mach number 1.8 (fig. 5(b)). This difficulty arises at Mach numbers below 2.0 because of both the increases in bleed mass flow and the additional reduction in captured mass flow due to subcritical matching. Figure 5(b) shows an advantage of $3\frac{1}{2}$ percent of ideal thrust at Mach number 1.48 when the inlet is sized at that Mach number for subcritical matching rather than for critical matching. However, this results in a loss of 2 percent of ideal thrust at Mach 3.0 compared with virtually no loss for critical-flow matching. Further optimization of this aspect is not fruitful without a specific application. The data do suggest, however, that varying the amount of bleed flow or ram-scoop geometry may alleviate the problem.

Furthermore, with respect to oversizing, the no-bleed effective-thrust-ratio curve is rather flat between Mach numbers of 2.5 and 1.4, whereas the bleed case still has a pronounced dip at Mach number 1.8. Comparison of the thrust loss due to pressure recovery at critical flow with that for peak recovery (which corresponds to a rubber or variable size inlet) shows that the difference is no more than 6.5 percent of ideal thrust, which is not enough to offset the drag due to operation at peak recovery. Therefore, the depressed regions of the effective-thrust curves primarily reflect the peaking of the drag curve. (As discussed previously, the high drag values are largely additive drag due to the choked-throat mass-flow-ratio restriction.) These regions of low effective thrust could decrease the aircraft thrust margin (effective thrust minus aircraft drag) sufficiently to seriously affect acceleration characteristics.

A further example of an off-design matching is a Mach 1.0 matched, no-bleed inlet with a fixed-spike position of 29.5° (the design $\theta_l$). Here also is the improved thrust margin due to pressure recovery below Mach 2.4; however, only below Mach 1.8 is this increased thrust able to overcome the oversizing spillage drag penalty as compared with the Mach 3.0 variable-spike match.

The penalty at the design Mach number of 3.0 due to oversizing is only 2 percent of ideal thrust for the no-bleed case, which required about 25-percent increase in inlet area. For the bleed case the penalty is about 4 percent of ideal thrust for about 62-percent size increase; however, because of the lack of data below Mach number 1.48, the necessity of such a large size cannot definitely be established.

## CONCLUDING REMARKS

By translating the spike so that the second oblique shock remained on the cowl lip, the airflow at critical conditions of the inlet closely matched that required by a turbojet between Mach numbers 3.0 and 2.4.

At lower Mach numbers the throat was choked, which reduced the captured mass flow and increased normal-shock additive drag. The total drag reached a peak value at about Mach number 1.3 of 3.4 to 3.7 times the Mach number 3.0 value. Because of the reduced mass flow due to throat choking, oversizing of the inlet was required in order to avoid supercritical matching, and at higher Mach numbers a bypass drag for the excess captured flow was accepted. This procedure at best produced a relatively flat effective-thrust curve in the choked-throat region with a small bypass-drag penalty at the design point.

By using a fixed ram-scoop boundary-layer bleed, the effective thrust was increased 20 percent at Mach number 3.0 and was better than the no-bleed inlet down to Mach number 2.0. At Mach numbers between 2.0 and 1.4 the bleed-system mass flow increased without any appreciable pressure-recovery benefit and hence required a relatively larger inlet with the attendant oversizing penalties.

For an external-compression inlet, the maximum throat area (which occurs at the telescoping joint of the centerbody) is determined by considerations of cowl-lip projected area (and contour) and rate of centerbody turning at the compression-surface shoulder. Once the shoulder turning rate has been maximized, the only way to increase throat area (excluding secondary systems such as variable-angle cones, etc.) is at the expense of cowl-lip projected area. This exchange of cowl drag for reduced additive drag, although offering some improvement in the below-design-speed region, compromises the performance at the design point because of increased cowl drag. Therefore, this dilemma appears to be inherent with this type of inlet. Additional considerations that complicate the below-design-speed performance are flow detachment of the second cone and cowl lip.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, October 22, 1959

## REFERENCES

1. Connors, James F., Wise, George A., and Lovell, J. Calvin: Investigation of Translating-Double-Cone Axisymmetric Inlets with Cowl Projected Areas 40 and 20 Percent of Maximum at Mach Numbers from 3.0 to 2.0. NACA RM E57C06, 1957.

2. Connors, James F., Lovell, J. Calvin, and Wise, George A.: Effects of Internal-Area Distribution, Spike Translation, and Throat Boundary-Layer Control on Performance of a Double-Cone Axisymmetric Inlet at Mach Numbers from 3.0 to 2.0. NACA RM E57F03, 1957.

3. Allen, John L., and Mitchell, Glenn A.: Performance of a Mach Number 3.0 Design Axisymmetric Double-Cone External-Compression Inlet in Mach Number Range 2.07 to 1.48. NASA MEMO 12-22-58E, 1959.

4. Davis, Owen H., and Mitchell, Glenn A.: Performance of a Mach number 3.0 Design Axisymmetric Double-Cone External-Compression Inlet in the the Mach Number Range 1.97 to 0.79. NASA TM X-154, 1960.

5. Allen, J. L., and Beke, Andrew: Performance Comparison at Supersonic Speeds of Inlets Spilling Excess Flow by Means of Bow Shock, Conical Shock, or Bypass. NACA RM E53H11, 1953.

6. Hearth, Donald P., and Connors, James F.: A Performance Analysis of Methods for Handling Excess Inlet Flow at Supersonic Speeds. NACA TN 4270, 1958.

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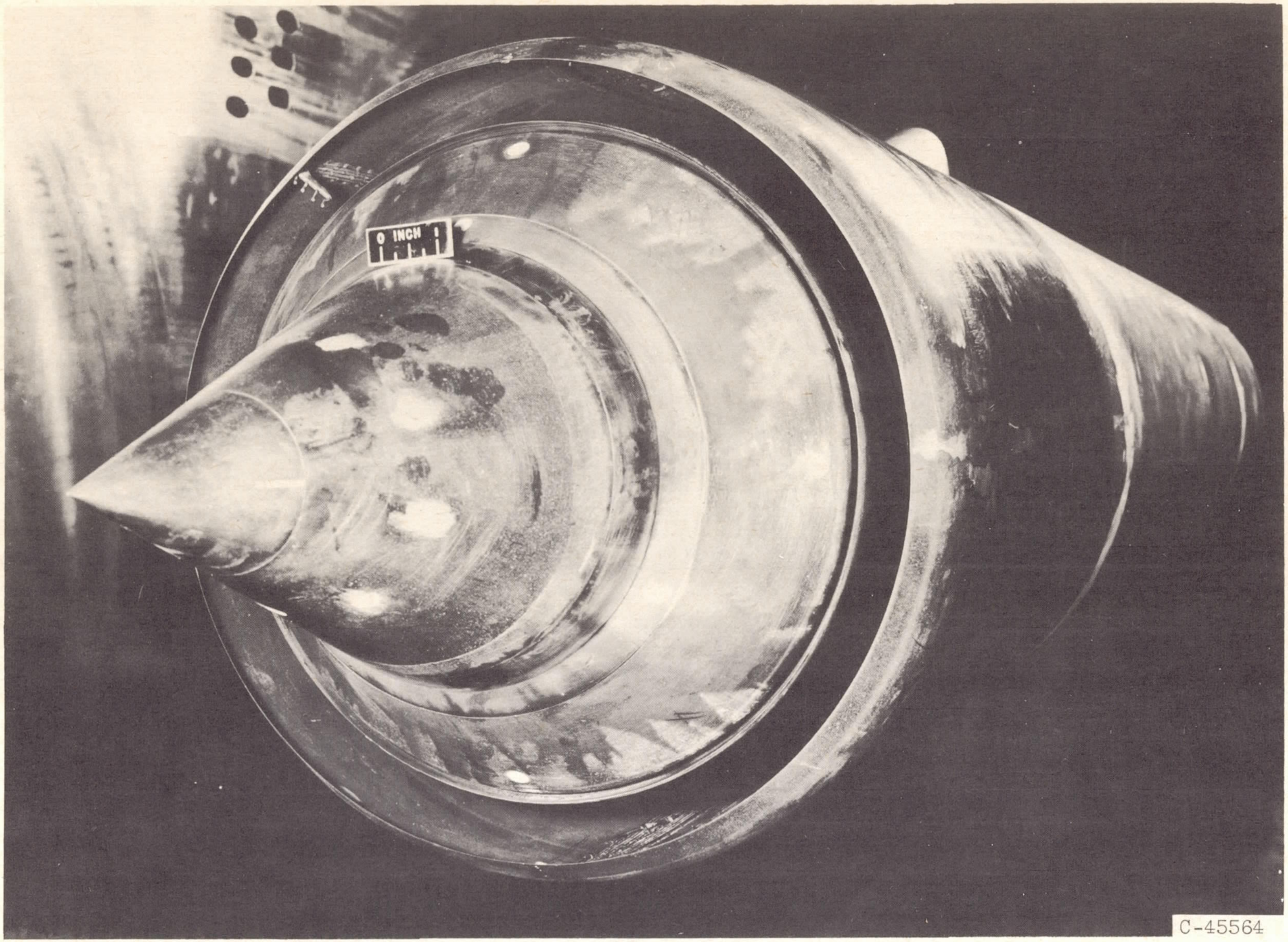


Figure 1. - Configuration with ram-scoop boundary-layer bleed.

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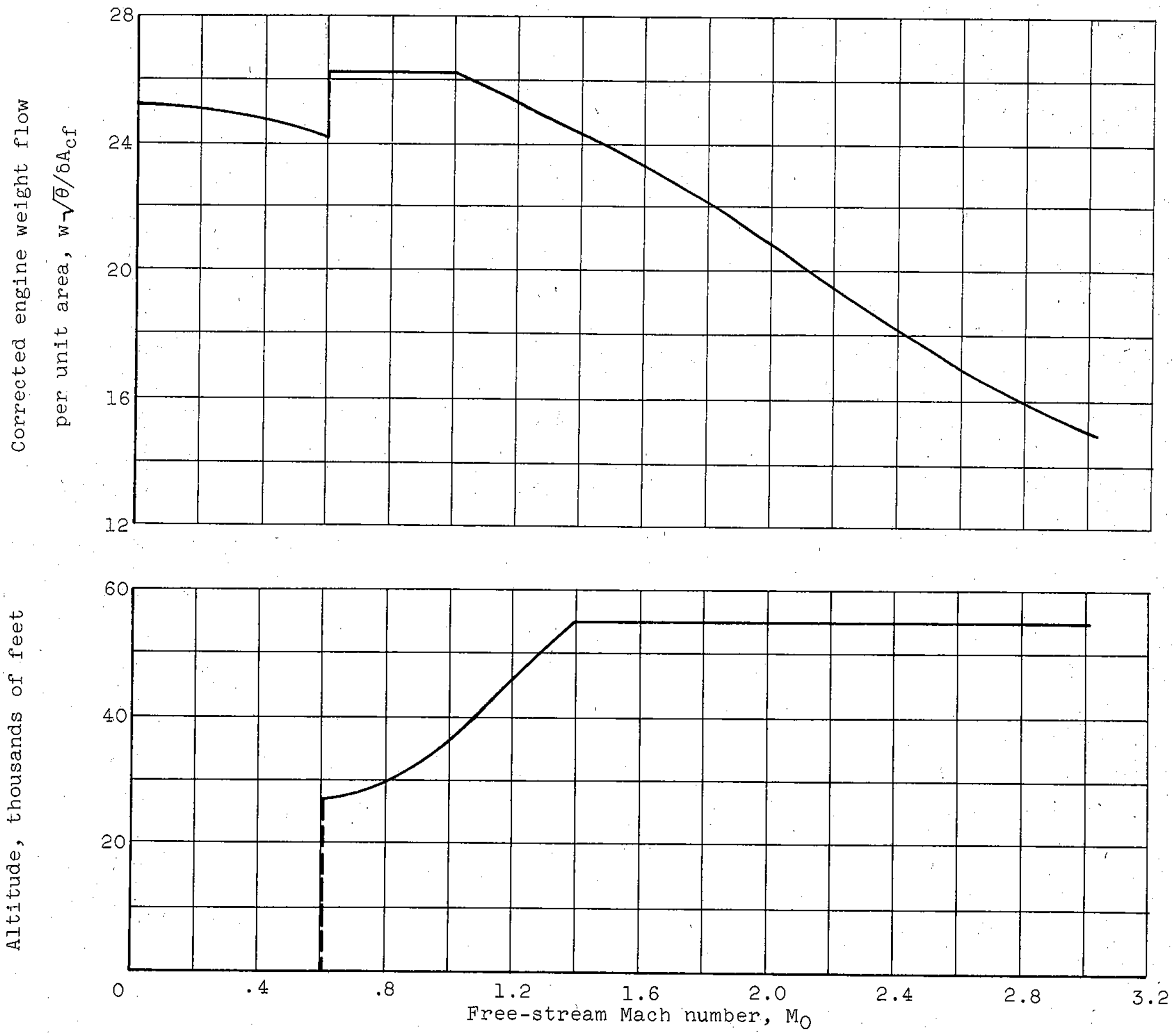


Figure 2. - Flight path and turbojet airflow schedule.

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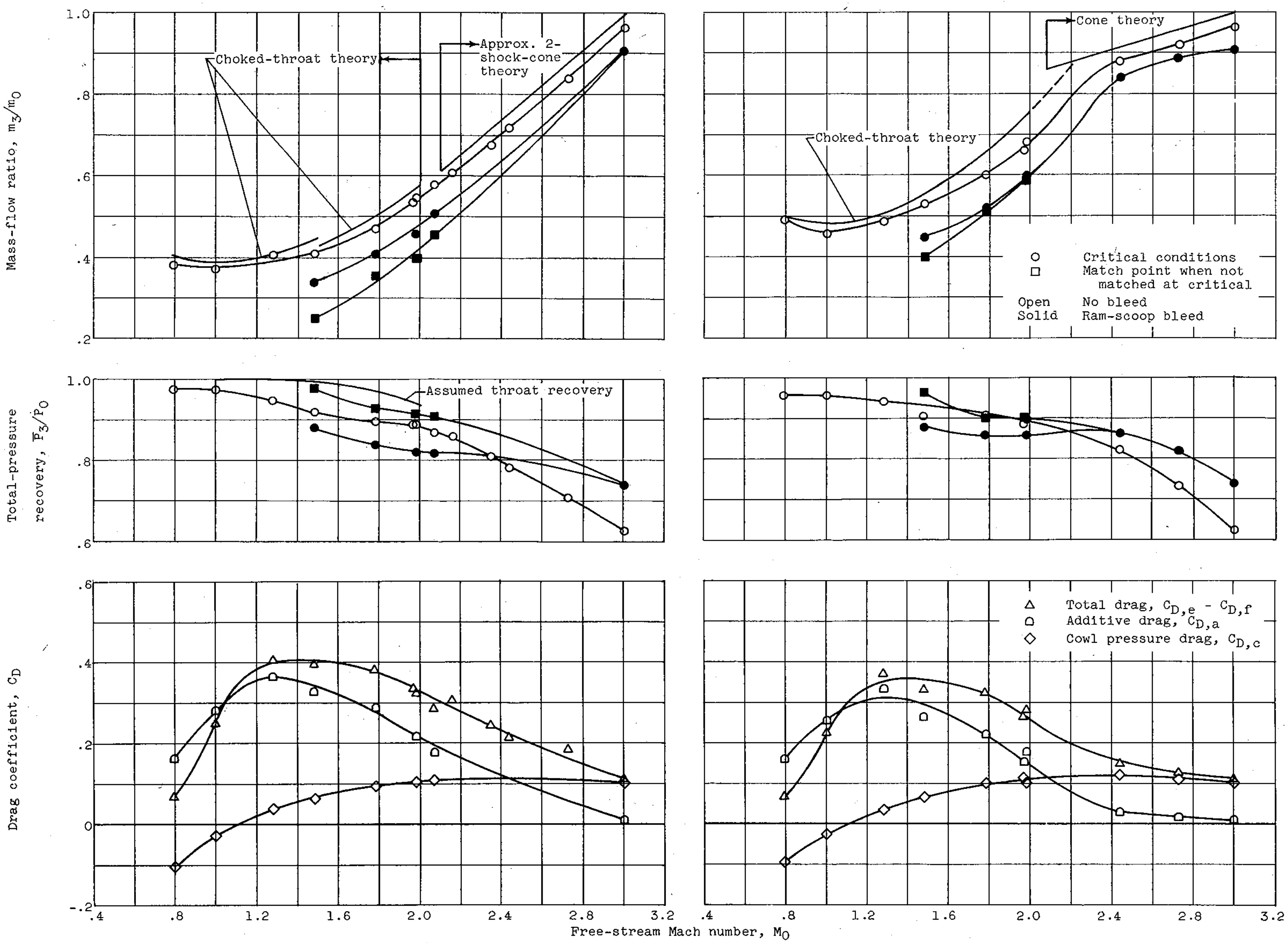


(a) Fixed spike.

(b) Translating spike.

Figure 3. - Basic inlet performance.

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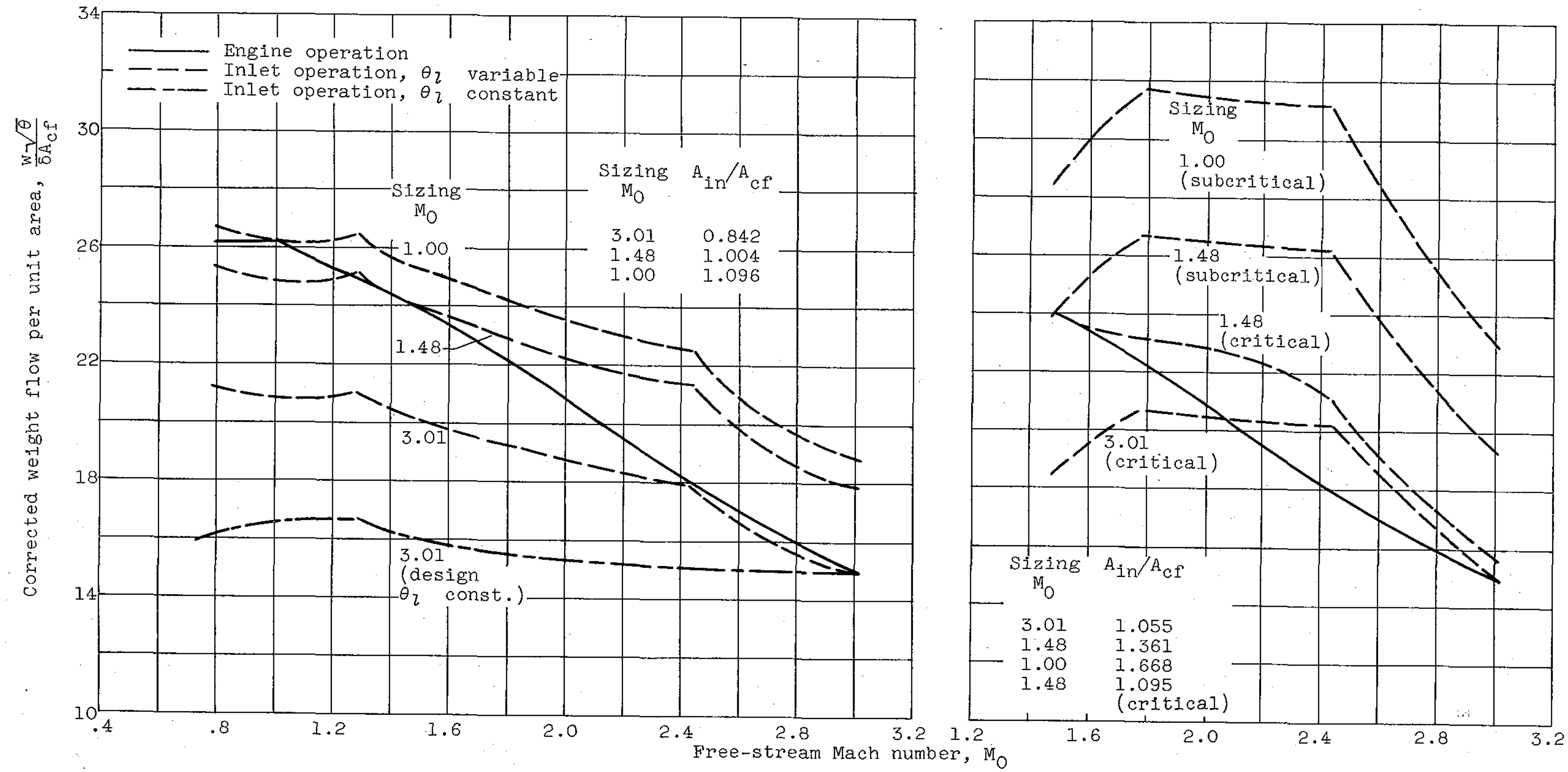


Figure 4. - Comparison of corrected weight flows of engine and various sizes of inlets. Scheduled $\theta_l$ except where noted.

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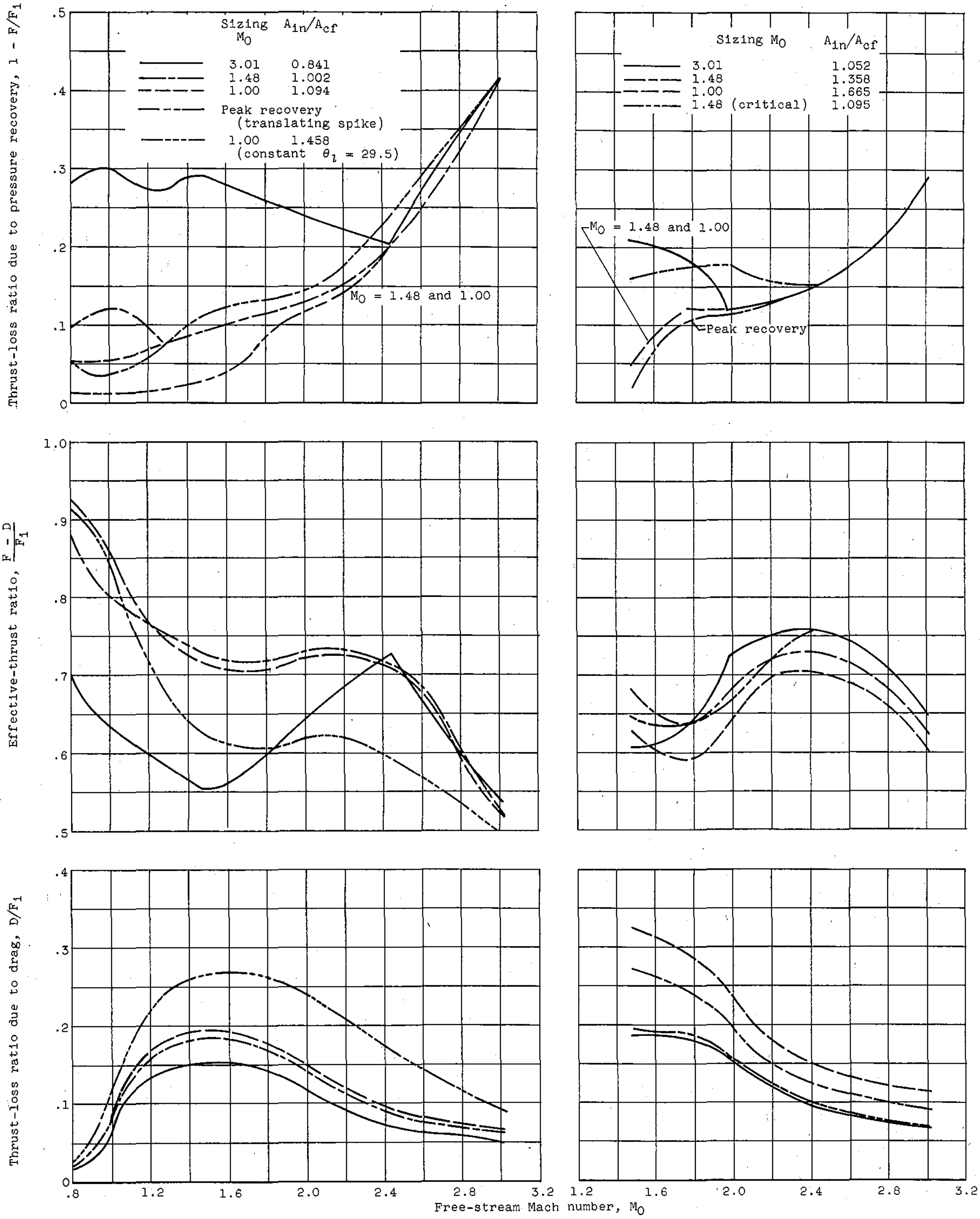


(a) No throat bleed.

(b) Throat bleed.

Figure 5. - Effective thrust and thrust-loss ratios for various sizes of inlets with varying spike position.